NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Technical Memorandum 33-746

Solid Rocket Technology Advancements for Space Tug and IUS Applications

W. Ascher
R. L. Bailey
J. W. Behm
W. Gin

(NASA-CR-145561) SOLID ROCKET TECHNOLOGY ADVANCEMENTS FOR SPACE TUG AND IUS APPLICATIONS (Jet Propulsion Lab.) 28 p HC $4.00 CSCL 21H
N76-11233
Unclas
G3/20 03090

JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

October 15, 1975

TECHNICAL REPORT STANDARD TITLE PAGE

1. Report No. 33-746	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle SOLID ROCKET TECHNOLOGY ADVANCEMENTS FOR SPACE TUG AND IUS APPLICATIONS		5. Report Date October 15, 1975
		6. Performing Organization Code
7. Author(s) W. Ascher, R. L. Bailey, J. W. Behm, W. Gin		8. Performing Organization Report No.
9. Performing Organization Name and Address JET PROPULSION LABORATORY California Institute of Technology 4800 Oak Grove Drive Pasadena, California 91103		10. Work Unit No.
		11. Contract or Grant No. NAS 7-100
		13. Type of Report and Period Covered Technical Memorandum
12. Sponsoring Agency Name and Address NATIONAL AERONAUTICS AND SPACE ADMINISTRATION Washington, D.C. 20546		14. Sponsoring Agency Code
15. Supplementary Notes		

16. Abstract

It presently appears that in order for the Shuttle Tug or Interim Upper Stage (IUS) to capture all the missions in the current mission model for the Tug and the IUS, an auxiliary or kick stage, probably using a solid propellant rocket motor, is required. The purpose of this paper is to present and review the two solid propellant rocket motor technology concepts being sponsored and pursued by the NASA Jet Propulsion Laboratory to meet the general requirements of the motor designs applicable to that auxiliary or kick stage. One concept, called the "Advanced Propulsion Module" motor, is an 1800-kg, high-mass-fraction motor, which is single-burn and contains Class 2 propellant. The other concept, called the "High Energy Upper Stage Restartable Solid," is a two-burn (stop-restartable on command) motor which at present contains 1400 kg of Class 7 propellant. The details and status of the motor design and component and motor test results to date are presented, along with the schedule for future work.

17. Key Words (Selected by Author(s))	18. Distribution Statement
Launch Vehicles and Space Vehicles Space Transportation Spacecraft Design, Testing and Performance Spacecraft Propulsion and Power	Unclassified -- Unlimited

19. Security Classif. (of this report)	20. Security Classif. (of this page)	21. No. of Pages	22. Price
Unclassified	Unclassified	27	

HOW TO FILL OUT THE TECHNICAL REPORT STANDARD TITLE PAGE

Make items 1, 4, 5, 9, 12, and 13 agree with the corresponding information on the report cover. Use all capital letters for title (item 4). Leave items 2, 6, and 14 blank. Complete the remaining items as follows:

3. Recipient's Catalog No. Reserved for use by report recipients.

7. Author(s). Include corresponding information from the report cover. In addition, list the affiliation of an author if it differs from that of the performing organization.

8. Performing Organization Report No. Insert if performing organization wishes to assign this number.

10. Work Unit No. Use the agency-wide code (for example, 923-50-10-06-72), which uniquely identifies the work unit under which the work was authorized. Non-NASA performing organizations will leave this blank.

11. Insert the number of the contract or grant under which the report was prepared.

15. Supplementary Notes. Enter information not included elsewhere but useful, such as: Prepared in cooperation with... Translation of (or by)... Presented at conference of... To be published in...

16. Abstract. Include a brief (not to exceed 200 words) factual summary of the most significant information contained in the report. If possible, the abstract of a classified report should be unclassified. If the report contains a significant bibliography or literature survey, mention it here.

17. Key Words. Insert terms or short phrases selected by the author that identify the principal subjects covered in the report, and that are sufficiently specific and precise to be used for cataloging.

18. Distribution Statement. Enter one of the authorized statements used to denote releasability to the public or a limitation on dissemination for reasons other than security of defense information. Authorized statements are "Unclassified—Unlimited," "U.S. Government and Contractors only," "U.S. Government Agencies only," and "NASA and NASA Contractors only."

19. Security Classification (of report). NOTE: Reports carrying a security classification will require additional markings giving security and downgrading information as specified by the Security Requirements Checklist and the DoD Industrial Security Manual (DoD 5220.22-M).

20. Security Classification (of this page). NOTE: Because this page may be used in preparing announcements, bibliographies, and data banks, it should be unclassified if possible. If a classification is required, indicate separately the classification of the title and the abstract by following these items with either "(U)" for unclassified, or "(C)" or "(S)" as applicable for classified items.

21. No. of Pages. Insert the number of pages.

22. Price. Insert the price set by the Clearinghouse for Federal Scientific and Technical Information or the Government Printing Office, if known.

PREFACE

The work described in this report was performed by the Propulsion Division of the Jet Propulsion Laboratory.

CONTENTS

ABSTRACT

It presently appears that in order for the Shuttle Tug or Interim Upper Stage (IUS) to capture all the missions in the current mission model for the Tug and the IUS, an auxiliary or kick stage, probably using a solid propellant rocket motor, is required. The purpose of this paper is to present and review the two solid propellant rocket motor technology concepts being sponsored and pursued by the NASA Jet Propulsion Laboratory to meet the general requirements of the motor designs applicable to that auxiliary or kick stage. One concept, called the "Advanced Propulsion Module" motor, is an 1800-kg, high-mass-fraction motor, which is single-burn and contains Class 2 propellant. The other concept, called the "High Energy Upper Stage Restartable Solid," is a two-burn (stop-restartable on command) motor which at present contains 1400 kg of Class 7 propellant. The details and status of the motor design and component and motor test results to date are presented, along with the schedule for future work.

INTRODUCTION

The objective of this paper is to present and review two solid propellant rocket motor technology projects being pursued by the NASA Jet Propulsion Laboratory to meet the energetic planetary and earth orbit unmanned spacecraft missions which will utilize some elements of the Space Transportation System. This motor technology work is addressed to the auxiliary or kick stage of the Space Tug and the Interim Upper Stage (IUS) of the Shuttle envisioned for applications in the 1980 decade. Planetary missions such as the Pioneer and Mariner class of outer planet spacecraft typically require large velocity increments. Earth orbit missions can range from low earth orbiters to geosynchronous and very high elliptical earth orbiters, with both the planetary missions and earth orbiters starting from Shuttle orbiter altitudes (see Fig. 1).

The paper will describe the current state of the art of solid propellant rocketry applicable to kick stage implementation in the Shuttle Tug or IUS and then proceed to discuss the technological advancements being

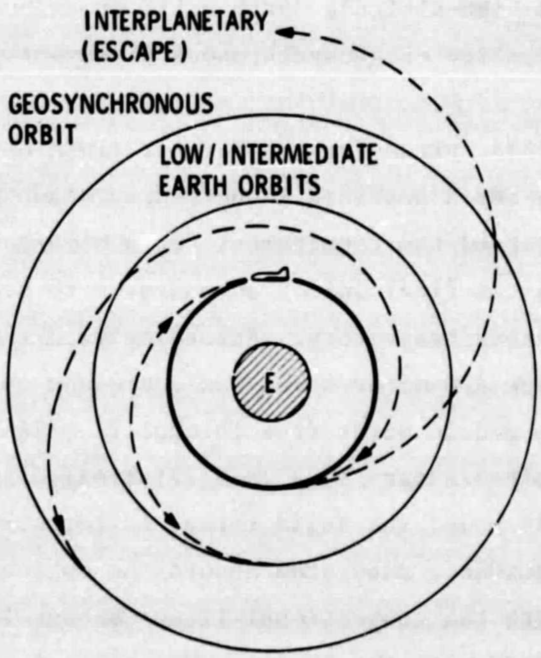


Fig. 1. Tug/IUS Auxiliary Stage Geocentric Propulsive Maneuvers

developed by NASA. These advancements revolve about two technology projects sponsored by the NASA Office of Aeronautics and Space Technology: the Advanced Propulsion Module (APM) motor, which is a high mass fraction, 1800-kg, single-burn, Class 2 propellant motor and the High Energy Upper Stage-Restartable Solid (HEUS-RS) motor, a two-burn motor with 1400 kg of Class 7 propellant. Details and status of the motor design and component and motor test results will be presented, along with the schedule for future work aimed at arriving at a timely state of technology readiness.

BACKGROUND

The traditional beneficial attributes of operational simplicity, low cost, and volumetric efficiency of solid rockets place them as prime candidates for upper stage augmentation of launch vehicles in order to meet the more energetic unmanned spacecraft mission requirements in a cost-effective manner. In 1968, recognizing this need, NASA/JPL initiated the HEUS-RS motor project, which was subcontracted to Hercules, Incorporated, for the demonstration of a high-energy (beryllium) propellant motor with a command capability for start-stop and restart-stop. This capability should enable an auxiliary stage of the Space Tug or IUS to perform a variety of low- and high-altitude earth orbit transfers of satellites, including the possibility of geosynchronous placement and retrieval.

More recently, in 1974, plans for the Mariner class of outer planet missions, specifically for a possible Titan/Centaur-launched Mariner Jupiter Orbiter 1981, identified the requirement for a high-performance, 1800-kg motor to be used in the final injection maneuver to send the spacecraft into its interplanetary trajectory. Accordingly, JPL started a technology project on the APM motor which was conceived as a technology and size upgrade of the module motor (the Thiokol TE-M-364-4) employed in the ongoing Mariner Jupiter/Saturn 1977 (MJS'77) mission in a powered spacecraft mode. In this mode, the solid rocket is thrust-vector and roll controlled by the guidance subsystem aboard the spacecraft. Although intended for use with the conventional Titan/Centaur launch vehicle, the same motor can be used with the Shuttle IUS or Tug for a Shuttle-launched outer planet spacecraft, because in both instances the solid rocket is

used for the final injection maneuver. A larger and more efficient motor than that presently used for MJS'77 can provide not only more payload capability but the potential for reduced trip time, thus reducing costs and enhancing mission reliability.

Several studies currently in progress have established the need for a solid rocket kick stage on the Space Tug and IUS designs to enable total mission model capture. (The planetary mission model has been examined in relation to IUS kick stage needs in Ref. 1.) These studies, in conjunction with certain programmatic decisions to be made in the near future, will define more specific or optimal sizes, operational capabilities, performance, envelope, and environmental requirements of the solid rocket motors.

THE ADVANCED PROPULSION MODULE MOTOR

Current Technology

To illustrate the application concept of the APM motor, a quick look at a current flight project is useful. The overall configuration of the spacecraft, integrated with the solid propulsion module (SPM), is shown in Fig. 2. The solid rocket motor (Fig. 3) is housed inside a 94-kg semi-monocoque aluminum structure and is bolted to a girth flange around the cylindrical section of the motor chamber. After earth orbit separation from the Titan IIIE/Centaur launch vehicle, the SPM motor imparts a velocity increment of 2100 m/s to the spacecraft. Thrust vector control is provided separately by a liquid monopropellant system. The propulsion module mass consists of solid propellant and two categories of inerts--motor inerts and module attachment structure. The ratio of inert mass to propellant mass, expressed as a percentage, is a measure of the state of technology. The values achieved by present technology are 8.06% for motor inerts and 9.05% for module attachment structure. The sum of 17.11% is an indicator of overall structure efficiency.

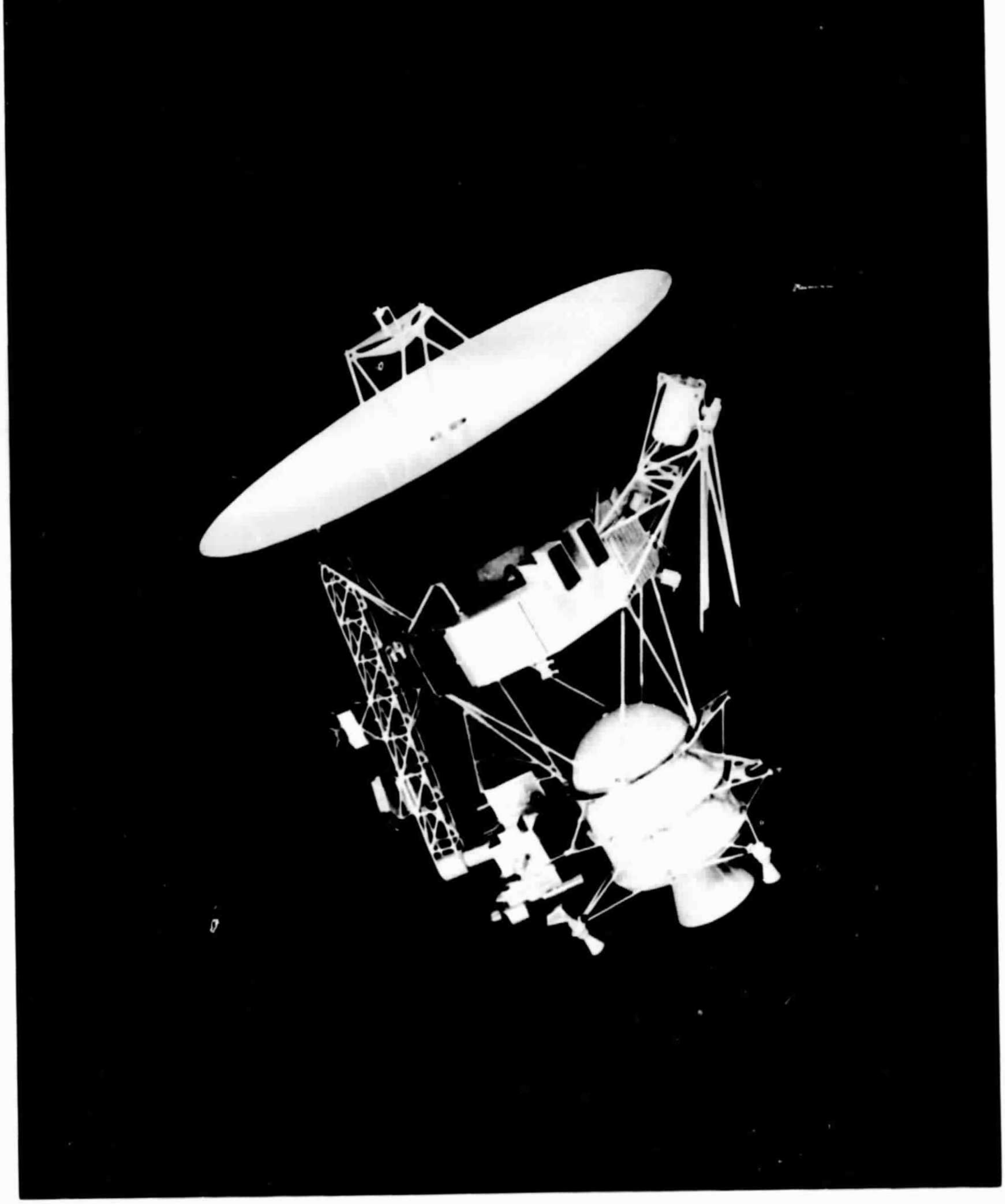
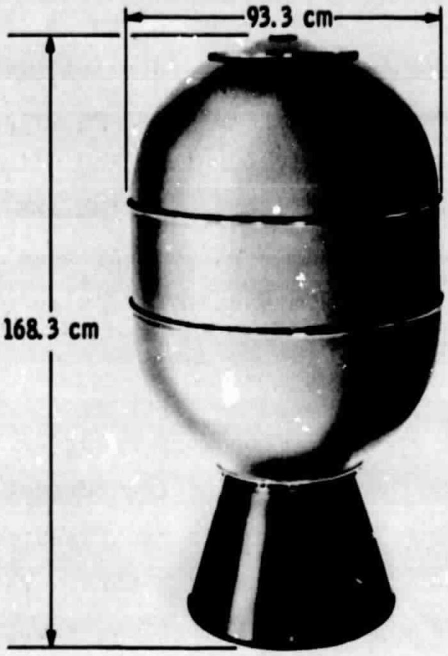


Fig. 2. MJS'77 Spacecraft



CHARACTERISTICS

AVERAGE THRUST, N	66,700
BURNING TIME, s	44
PROPELLANT MASS, kg	1039
MASS FRACTION	0.925
SPECIFIC IMPULSE, m/s	2800
PROPELLANT TYPE	CTPB/AP/Al
DESIGNATION	TE-M-364-4

Fig. 3. MJS'77 Propulsion Module Solid Propellant Motor

Note that the motor inert mass is about the same as the module attachment structure mass. This highlights an important aspect for technology improvement separate from that of motors--the reduction of module attachment structure mass.

In examining the features of current technology and then comparing them to their advanced-technology counterparts, for a larger spacecraft payload, it is useful to have a common size reference, defined as follows:

Motor mass, kg	1800
Average thrust, kN	66.7
Maximum thrust, kN	80.1
Nozzle expansion ratio	80

The motor size selection was based on an early analysis of a Mariner Jupiter Orbiter 1981 mission, with a 1360-kg spacecraft, and use of the Titan IIIE/Centaur launch vehicle. It was noted then that if the motor for a solid propulsion module is to be kept within a reasonable mass bracket (1750 to 2100 kg), current-technology propellant mass fractions

would require delivered average vacuum specific impulse values in the region of 3040 m/s, well in excess of values attainable in existing Class 2 solid propellants (2880 m/s). In examining the relationships between motor mass, propellant mass fraction, and specific impulse, and with parallel design and sizing studies, the target criteria for the APM motor size were selected to be a mass of 1800 kg, a propellant mass fraction of 0.95, and an average delivered vacuum specific impulse of 2913 m/s.

To arrive at the characteristics of the current-technology motor, a point design was carried out, scaling up the features of the current MJS'77 SPM motor. The average and maximum thrust levels are fixed at the same values as those of the SPM motor for the larger motors. The thrust level constraint was imposed by a system consideration, i.e., compatibility with the liquid monopropellant thrust vector control (TVC) system developed for the ongoing MJS'77 spacecraft.[2] The four existing 445-N thrusters could be used; probably the only system change needed would be added monopropellant tankage for TVC during a longer burn.

An overview of current and advanced technology features for comparison purposes is presented in Table 1. The column on the left describes the components of the TE-M-364-4 motor,[3] and applies also to the scale-up of this design to the 1800-kg size.

The effect of improvements in inert hardware technology is best illustrated in steps. As a reference, the 1800-kg current-technology motor, scaled up from the 1122-kg MJS'77 SPM motor to meet the new requirements stated earlier, exhibits a propellant mass fraction of 0.923. This reference point is shown at the bottom left of Fig. 4. As individual inert technology improvements (described in Table 1) are introduced, the effect on propellant mass fraction is shown in the remaining parts of Fig. 4. The marked inert mass reductions resulting from the introduction of advanced component materials and fabrication methods illustrated by these specific point designs are based on testing at JPL and/or other government and industry installations and design margins to minimize risk.

Table 1

TECHNOLOGY FEATURES

COMPONENT	CURRENT TECHNOLOGY	ADVANCED TECHNOLOGY
Chamber	Titanium 6Al-4V	Kevlar-49, filament-wound with epoxy-base resin
Nozzle	Glass-, silica-, asbestos-, and carbon-reinforced phenolic ablatives	Carbon-carbon exit cone; carbon-reinforced phenolic support structure; carbon felt radiation shield (S.G. = 0.032)
Insulation	Silica-loaded ethylene-propylene elastomer (S.G. = 1.1). Vulcanized after layup	Silicone elastomer loaded with phenolic microballoons (S.G. = 0.66), cast in place, cured at low temperature, machine-trimmed
Liner	Carboxyl-terminated polybutadiene	None
Igniter/Safety and Arming Device	Existing TE-M-364-4 pyrogen system	Existing TE-M-364-4 pyrogen system
Propellant	Class 2, carboxyl-terminated polybutadiene/ammonium perchlorate/aluminum, 86% solids loading (16% aluminum, S.G. = 1.738)	Class 2, hydroxyl-terminated polybutadiene/ammonium perchlorate/aluminum, 86% solids loading (18% aluminum, S.G. = 1.758)

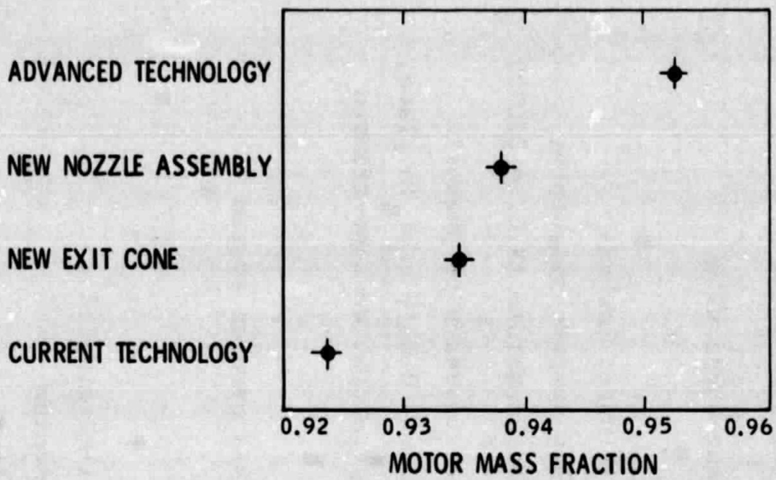


Fig. 4. Effects of Technology on Solid Motor Mass Fraction

Solid propellants used in connection with the Shuttle are presently required to be Class 2. This is essentially a low-hazard classification, stipulated by the Department of Transportation, determined by a series of tests: when subjected to various stimuli, such as friction, impact, and sparking, the solid propellant will not detonate. Launch vehicles such as Titan IIIE/Centaur generally do not restrict the introduction of Class 7 solid propellants. In contrast with the Class 2 hazard, Class 7 solid propellants are high-hazard, subject to detonation under certain conditions.

Perhaps the largest reduction in program risk and cost is connected with the specific selection of the Class 2 solid propellant for the APM motor. This is an 86 weight % solids-loaded hydroxyl-terminated polybutadiene (HTPB)/ammonium perchlorate/aluminum formulation, containing 18 weight % aluminum. The formulation is currently undergoing evaluation of physical and ballistic properties. The propellant exhibits excellent physical properties, has burn rate characteristics suitable for the interior ballistic configuration design and performance needed in the APM motor, and is compatible with (bondable to) the advanced interior insulation material selected for the motor design. The need for a liner material between the propellant and the insulation has been eliminated by the

use of the JPL-developed technique of rinsing the interior insulation surface with toluene diisocyanate (TDI) prior to propellant casting and case-bonding. Separate experiments have shown that increased solids loading (from the current 86% to 90%) will result in higher values of specific impulse and density than those currently needed. This growth capability is an important reserve for future performance needs.

The propellant used in the current-technology SPM motor for MJS'77 is also Class 2, and is an 86 weight % solids-loaded carboxyl-terminated polybutadiene (CTPB)/ammonium perchlorate/aluminum formulation, containing 16 weight % aluminum. With the low nozzle expansion ratio (30.8) of that motor, the average delivered vacuum specific impulse is 2805 m/s. If the ablative nozzle design is modified to have the same expansion ratio (80) as the APM motor, the predicted average delivered vacuum specific impulse is 2879 m/s. By introducing an all carbon-carbon nozzle exit cone, the specific impulse prediction is increased to 2908 m/s. The difference between these two values is due to anticipated differences in nozzle roughness and erosion during firing. This is based on the results of tests conducted at the United Technology Center[4] on identical motors with identical 10.4-cm-diameter throats. One set of nozzle exit cones was made of carbon-carbon material, while the other set, of identical interior geometry, was made of carbon and silica-phenolic material.

Results of studies made at JPL indicate that for typical applications of the APM motor to Shuttle Tug or IUS missions, the sensitivity of payload weight to specific impulse is significantly lower than the sensitivity to propulsion module mass fraction (the ratio of propellant mass to the propulsion module mass). Hence, the improvement effort in technology places a heavy emphasis on inert weight reduction, without an attendant change in specific impulse.

Motor Details and Status

The configuration of the APM motor is shown in Fig. 5. Performance and weight characteristics are provided in Fig. 6. A sizing study

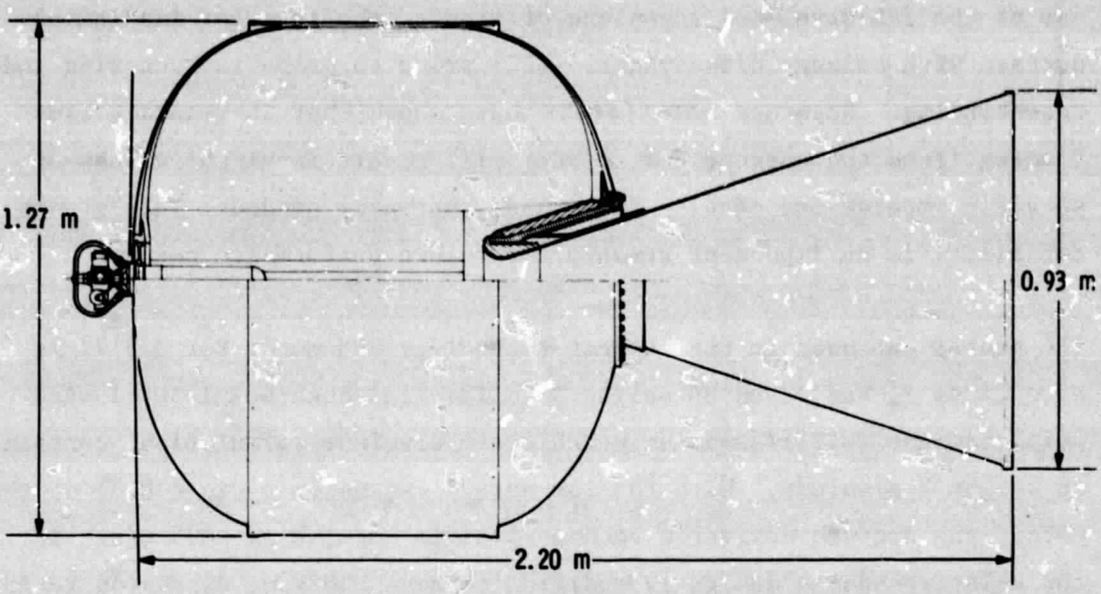
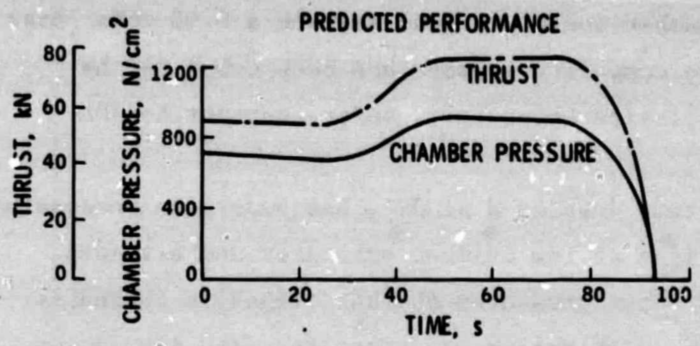


Fig. 5. APM Motor Configuration

established the fact that this APM motor design is scalable with certainty over the range of masses from 1100 to 7750 kg. Hence, once the APM motor technology is demonstrated in the current 1800-kg size, the results are directly applicable to a wide range of motor sizes. Another degree of flexibility is achieved with this motor design: The propellant configuration, masses, and performance shown are for a design which has about 16% more propellant than for the nominal 1800-kg motor, resulting in a motor mass of 2069 kg. The propellant charge design can be off-loaded readily to meet the nominal 1800-kg motor mass. This flexibility permits the configuration of a family of propellant charges to provide a variety of thrust vs. time histories tailored to meet different acceleration and acceleration rate constraints of various applications.

The current-technology MJS'77 motor does not contribute to the load-carrying function of the module structure. The APM motor configuration, however, is significantly different in that the cylindrical section of the chamber and skirts is designed as a primary load-carrying element of the propulsion module structure. If the current MJS'77 structure were scaled up to fit the 1800-kg size of a scaled-up current-technology



	MASS, kg
CHAMBER	33.32
NOZZLE	24.00
INSULATION	21.63
IGNITER	3.40
BOLTS	0.85
INERTS	83.20
PROPELLANT	1985.80
MOTOR	2069.00*
PROPELLANT MASS FRACTION	0.960*

*CAN BE OFF-LOADED TO 1800 kg, WITH PMF = 0.954

Fig. 6. APM Motor Performance and Mass

motor, its mass would be 151 kg. If advantage is taken of significant progress in composite tubular and honeycomb structural elements, the module structure mass for the APM motor is estimated to be about 51 kg.

On this basis, one can now compare propulsion module mass fractions for current and advanced-technology configurations, both for the nominal 1800-kg motor mass:

	Propulsion Module Mass Fraction
Scaled-up current technology	0.851
Advanced technology	0.924

Component Details and Status

The chamber is a 1.27-m-diameter by 1.20-m-long composite filament-wound structure. The material system is Kevlar-49 (Dupont) filament

and HBRF-55A resin selected on the basis of weight, strength, and cost. The forward and aft opening reinforcements are made of 7075 T-73 aluminum alloy. In comparison, other composites, such as S-901 glass, or homogeneous metals, such as Titanium 6 Al-4V, appear less desirable.[5] Indeed, the weight of a titanium chamber would preclude meeting a 0.95 motor mass fraction. The design of the composite chamber has been completed by Hercules Inc., Allegeny Ballistics Laboratory, under contract to JPL.

The interior insulation system, developed at JPL, has undergone successful demonstrations in motor firings at low chamber pressures and extended burn times. Testing at APM motor pressures and burn times is currently under way in subscale motors, with encouraging results. The insulation, designated SIPB, is a castable combination of room-temperature-vulcanizing silicone elastomer (General Electric RTV-615) and Bakelite phenolic microballoons (Union Carbide). The specific gravity of this SIPB system is 0.66, as compared with a specific gravity of 1.1 for a silica- and asbestos-loaded rubber of equivalent insulative and erosion performance (for example, a loaded ethylene-propylene terpolymer elastomer, Hilgard V-4030). Additional significant advantages of the SIPB system are the simplification of installation in the motor and attendant reduction of risk and cost. The SIPB system is cast onto the interior of the motor chamber after that surface is primed, cured at 66°C overnight, and finally trimmed to a continuous contour with an internal lathe. In contrast, rubber systems such as V-4030 come in calandered unvulcanized sheets (0.9 and 2 mm thick); after priming, pattern pieces are applied to the interior chamber surface in a series of stepped layers, approximating the desired continuous contour. The chamber and insulation are then vacuum-bagged, and the insulation typically is vulcanized in an autoclave at 149°C for 2 h.

The strain capability of the propellant and insulation, and the bond strengths at the chamber-to-insulation and insulation-to-propellant interfaces are currently being evaluated.

The APM motor nozzle design (see Fig. 5) is a fixed configuration, markedly lower in weight than an equivalent current-technology ablative

materials nozzle. The configuration shown is an 18-deg half-angle cone with an 80:1 expansion ratio. About 25% of the nozzle length is submerged within the motor chamber. Note that total motor length is 2.2 m, of which 1 m is the external part of the nozzle.

Except for the ring bolt flange adapter subassembly made of 7075-T73 aluminum alloy, all nozzle parts are nonmetallic. The reentrant support structure is a carbon-phenolic tape layup, molded in the shape of a thick-walled cone frustum. Except for heat transferred by conduction through the threaded joint with the nozzle insert, the temperature of most of this structure, protected by SIPB insulation, is expected to remain low (93°C) at the end of pressurized motor operation. The principal loading of this cone structure is external pressure (motor internal pressure). The nozzle insert shown is made from high-density graphite (Graph-i-tite), with machined threads. The nozzle exit cone is a one-piece repyrolized carbon-carbon composite structure, with a specific gravity of 1.6. More complex carbon-carbon cone configurations, somewhat smaller in overall size, have been successfully built (by the Reflective Laminates Division of Fansteel Corporation) and test fired at simulated altitude in connection with JPL's low-thrust motor technology program.[6,7] The exit cone subassembly is threaded to the aft end of the nozzle throat insert. The joint location was selected to result in almost balanced axial load on the threads. Both sides of the annular gap between the exterior of the carbon-carbon exit cone and the interior of the carbon-phenolic support structure are covered by a 1-cm-thick layer of carbon felt, acting as a radiation barrier. The mass of this carbon felt (specific gravity of 0.032) is negligible. The major part of the carbon-carbon exit cone is 1.9 mm thick. Except for SIPB insulation, the overall nozzle design is almost entirely insensitive to burn time because it has a hot-running exit cone which does not ablate. Near the attachment point, exit cone temperatures are expected to stablize at 1100 to 1650°C.

Separate experiments cited earlier in the text have shown that, in addition to the obvious mass benefit, the non-ablating cone surface results in lower nozzle flow losses than for ablative nozzles with the

same initial geometry. Earlier tests with carbon-carbon exit cones have also shown capability for multiple firing exposure and great resistance to severe thermal shock (water impingement). For module integration, the nozzle configuration chosen is highly submerged to minimize installed motor length. A less reentrant nozzle would weigh less, have lower flow losses, and would permit a smaller chamber aft opening and hence a reduction in chamber mass. Installation design trade studies, including the counteracting increases in module structure mass, are needed once specific applications are identified.

The ignition and safe arm system shown in Fig. 5 is identical to the fully developed and flight-proven component currently in use on the TE-M-364-4 motor[3]. In the interest of minimizing risk and cost, no significant change from this selection is anticipated.

In selecting certain features of the overall motor design, one additional degree of flexibility has been considered. The aft opening of the chamber was chosen to be somewhat larger than needed for a fixed nozzle design. This was done to permit the incorporation of an omniaxial movable nozzle to provide thrust vector control. Studies have been initiated to identify thrust vector control requirements beyond MJS'77.

THE HIGH ENERGY UPPER STAGE RESTARTABLE SOLID MOTOR

Present Technology

Pre-Shuttle technology relating to solid propellant upper stage propulsion has generally been limited to single-burn configurations. In some instances, where an intermediate earth orbit is required (e.g., Tiros), a two-burn second stage (e.g., Delta-liquid) may be capable of achieving the final transfer burn to apogee. In most cases, the spacecraft or satellite carries a separate single-burn solid motor to accomplish the final spacecraft apogee maneuver.

The Scout is an example of an all-solid rocket launch vehicle with single-burn stages. Alternately, the Delta utilizes liquid propellant lower

stages, with a single-burn solid rocket providing the final third-stage impulse. The Minuteman and Poseidon are other examples of traditional staged single-burn solid rocket vehicles.

The BIIA*, although generally referred to as a single stage, actually is a two-stage propulsion configuration which incorporates two separate single-burn rocket motors. After completion of the first burn, the spent lower rocket motor is ejected some time prior to ignition of the second (upper) rocket motor. This scheme provides the advantage of being able to coast between burns but is constrained to utilizing two fixed impulses regardless of varying payload weights and/or differing orbit needs.

Future Technology

Future multi-burn upper-stage propulsion applications should be directed primarily at earth-orbital missions, where a flexible, "on-command," multiple-burn capability provides maximum mission capability through performance gains and mission versatility. The HEUS-RS is intended for use on a standard, "workhorse" stage capable of meeting many missions with a single solid rocket motor configuration. Figure 7 illustrates the performance advantages of a two-burn BII(2300) upper stage (including quench weight penalties) over a single-burn version using the same rocket motor. A single rocket with two-burn capability could be used conceptually in a great number of different mission modes.

The HEUS-RS demonstration program was originally conceived prior to the current Shuttle launch concept. As such, a number of studies were conducted which consistently indicated the HEUS-RS to be effective in improving baseline launch vehicle performance capabilities in a cost-effective manner.[8,9]

Looking into the future, it would appear that the favorable applicability of the HEUS-RS to conventional launch vehicles will generally carry over

---

*Boeing Burner IIA stage. First-burn impulse is provided by a TE-M-364-2 burn-to-depletion rocket motor and second-burn impulse by a separate TE-M-441 solid motor.

into the Shuttle/IUS Tug configurations. A somewhat limited examination has already been made with favorable results.[10] It may be more cost-effective to utilize low-cost, expendable, solid upper stages to place small or intermediate size payloads into low and intermediate earth orbits during the early Shuttle/IUS operations than to accomplish the same mission using a nonrecoverable IUS. Also, use of a low-cost, expendable solid upper stage with flexible operating characteristics could augment the IUS in a cost-effective manner whether or not the IUS is recoverable. Finally, a nonrecoverable solid upper stage could be utilized to reduce the total direct Tug energy requirements to facilitate Tug recovery.

HEUS-RS Technology

There is a great variety of propulsion schemes to choose from which can provide a multiple-burn capability. Centaur and the second stage of Delta are examples of biliquid propellant systems (cryogenic and earth-storable, respectively) of a two-burn capability. However, these designs are large, sophisticated, and relatively costly. Other multi-burn propulsion schemes, more related to solids, include hybrids, dual-chamber, and various mass augmentation concepts. While these schemes have all been proven feasible, relatively high cost, high complexity, and/or lower performance have resulted in only a few such configurations being flown, and then to a limited extent. Several other solid rocket schemes have received significant attention from a demonstration aspect or have been used in an operational system. The pintle nozzle is one approach to utilizing rapid depressurization characteristics of certain solid propellant formulations to terminate the combustion process in order to achieve on-command thrust termination. This system has the potential for many burns but is somewhat complex and would generally be more costly than more conventional solids. The pintle must survive very severe thermal environments inasmuch as it is located in the throat area of the nozzle. The wafer motor is an example of a pure solid rocket motor design with multi-burn capability but is somewhat less flexible because each impulse is fixed in size. A two-burn (boost/sustain) wafer motor design is currently being successfully flown as primary propulsion on board the SRAM missile.

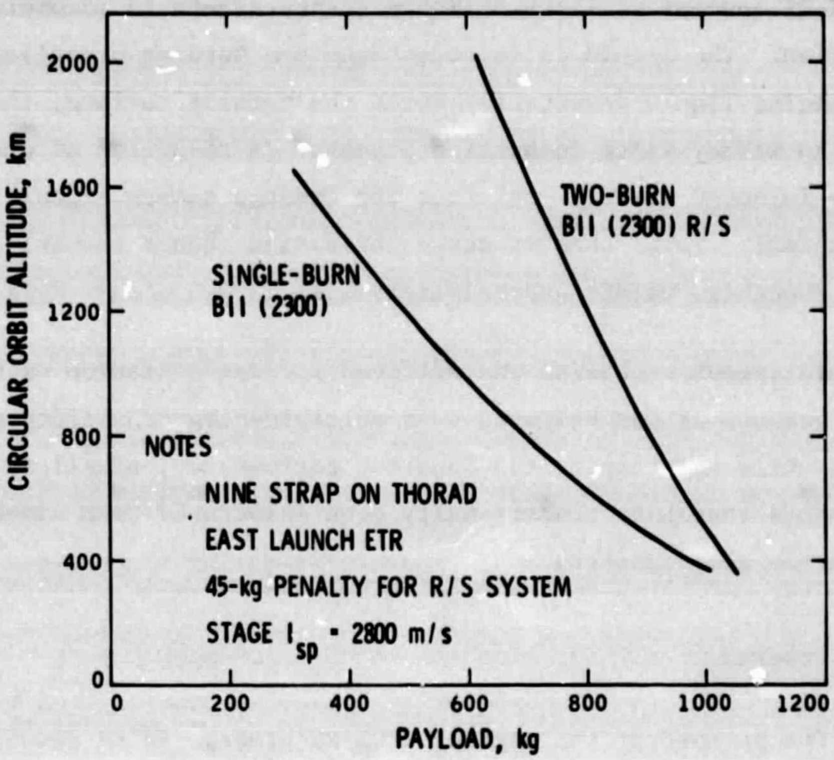


Fig. 7. BII (2300) Restartable Stage Comparison

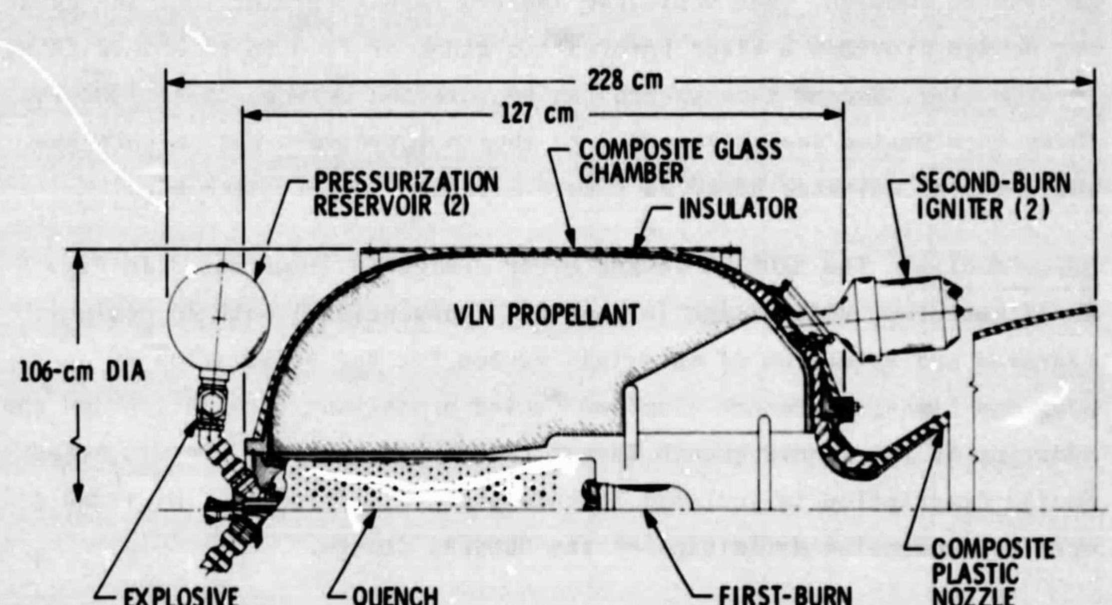


Fig. 8. HEUS-RS Motor Design

The HEUS-RS concept utilizes a liquid quench scheme to accomplish thrust termination. The liquid is injected onto the burning propellant surface. The impinging liquid momentarily cools the burning surface, thus terminating the steady-state combustion process. A reduction of chamber pressure is noted within 4 ms after the command quench signal, without thrust spikes. Total chamber depressurization (hence thrust) is accomplished smoothly, within approximately 1 s.

The liquid quench mechanism was selected for demonstration on the HEUS-RS program because it was believed that multiple-burn capability could be realized while maintaining the inherent performance, simplicity, and low cost aspects that have traditionally been associated with single-burn-to-depletion solid rockets.

HEUS-RS Program

Scope. The purpose of the ongoing HEUS-RS program is to demonstrate the feasibility of liquid quench with a 1400-kg propellant grain, contained in flight-type hardware, under simulated altitude back pressure conditions. A two-burn, two-termination capability is being demonstrated which utilizes a liquid quench concept to interrupt the propellant combustion process on command, thus achieving desired thrust termination. The present design provides a first termination range of from 66 to 85% of total impulse (I_T). Second termination can be selected between 85 to 100% I_T. These termination design ranges were chosen for demonstration purposes and could be adjusted based on future mission model requirements.

Motor Design. The HEUS-RS rocket motor design is illustrated in Fig. 8. As illustrated, this design is generally conventional both in design features and selection of materials except for the utilization of an advanced high-performance aluminum-fueled propellant formulation and the addition of the liquid quench thrust termination system. Only a brief design description is included in this paper. Reference 11 provides a more comprehensive definition of the HEUS-RS design.

The chamber is a filament-wound design using S904 fiberglass impregnated with ERL 2256/Tonox 6040 epoxy resin/curing agent.

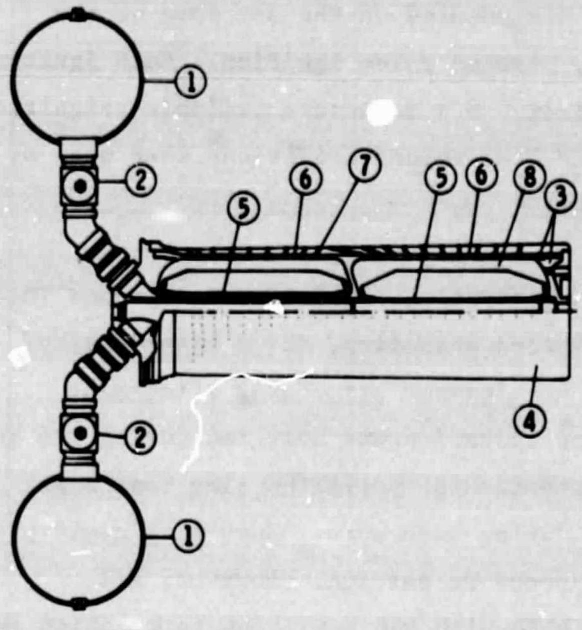


Fig. 9. Liquid HEUS-RS Quench System Diagram

The forward half of the chamber insulator is made of silica-filled styrene butadiene rubber (SBR). The aft insulator is fabricated from asbestos-filled SBR. Each insulator incorporates relief features to minimize residual grain stresses generated during post-cure cooldown.

The current propellant formulation consists of a high-performance (Class 7) crosslinked double-base (XLDB) formulation, designated as VLN, which contains aluminum. Earlier work utilized a XLBD formulation containing beryllium powder to maximize performance. The beryllium-fueled formulation was dropped because of environmental concerns caused by beryllium oxide being exhausted during ground testing operations.

The first-burn igniter, mounted on the aft end of the quench bottle, consists of a standard pyrotechnic design. The output charge is made up of boron/potassium nitrate pellets augmented by magnesium/teflon pellets. Redundant squib/detonator initiators, located at the forward end of the motor, initiate a mild detonating fuse which forms a portion of the ignition train and ignites the main charge pellets contained in the igniter basket.

Redundant pyrogen-type igniters are mounted on the aft dome of the chamber to achieve, "on command," second pulse ignition. Each igniter discharge duration is approximately 0.8 s to ensure reliable reignition of the residual charge under high free volume conditions when most of the original charge is burned out.

The nozzle assembly represents a conventional ablative design and incorporates (1) heat-resistant reinforced phenolics, (2) a high-density graphite throat insert, and (3) an aluminum structural attachment closure. The external nozzle configuration was selected during the initial design phase to minimize retention of beryllium slag that might have been generated and trapped during each burn. Such slag could provide a possible source of spontaneous reignition. However, all subsequent test firing results, both with the beryllium formulation and, more recently, using the current aluminized formulation, have demonstrated that metallic slag does not accumulate within the chamber. One of the major requirements of the nozzle is to reliably survive two-burn operation.

The liquid quench thrust termination system represents one of the unique aspects of the HEUS-RS motor design. The specific mechanization configuration was selected after examining alternate design approaches.[12,13,14] Figure 9 identifies key elements of the Hercules design, Fig. 10 illustrates the theoretical spray pattern relative to the burning surfaces, and Fig. 11 provides high-speed film coverage of the pattern quality provided by the actual full-scale hardware during ambient ground tests. The quench assembly tested was subsequently used to successfully quench the full-scale Phase II motor, which contained VID (beryllium) propellant.[15]

Quench Test Results. Table 2 summarizes quench design conditions and test firing results of all demonstrations conducted to date. All tests to date have been positive regarding successful initial termination.

The small motor tests were performed to verify the effectiveness of the quench parameters represented by the full-scale design. These small

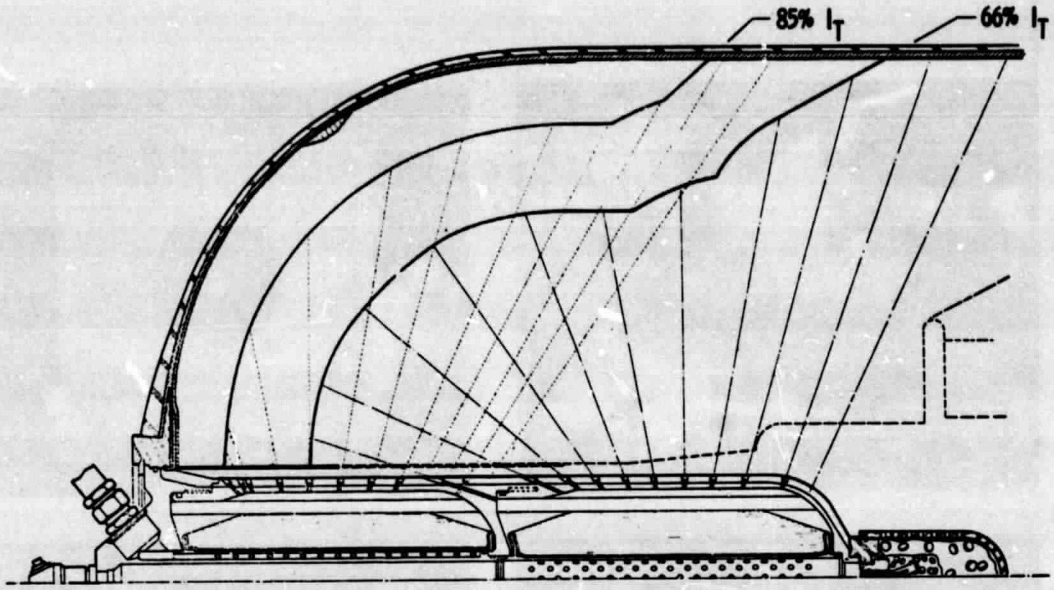


Fig. 10. Quench Impingement Patterns

motors were exhausted into a large vacuum tank in order to more adequately simulate the applicable back pressure encountered in the hard vacuum of space. Because of the relatively low pressure deflagration limit (the pressure below which sustained combustion cannot be maintained for a given propellant) associated with propellant formulations of interest, near-vacuum back pressure test conditions are required to demonstrate that the propellant will not inadvertently reignite spontaneously. All of the small motors were quenched successfully and did not experience spontaneous reignition. As Table 2 indicates, large motors tested at near-sea-level conditions can be quenched successfully, but spontaneous reignition will occur (unless artificially cooled) because the propellant is exposed to ambient pressures (above its pressure deflagration) limit while being heated above its autoignition temperature by heat soak back from hot elements of the quench motor. References 11 and 16 more fully document HEUS-RS quench test experience compiled to date.

Performance. Table 3 provides performance estimates for two classes of flight weight HEUS-RS designs. The top design represents a light-

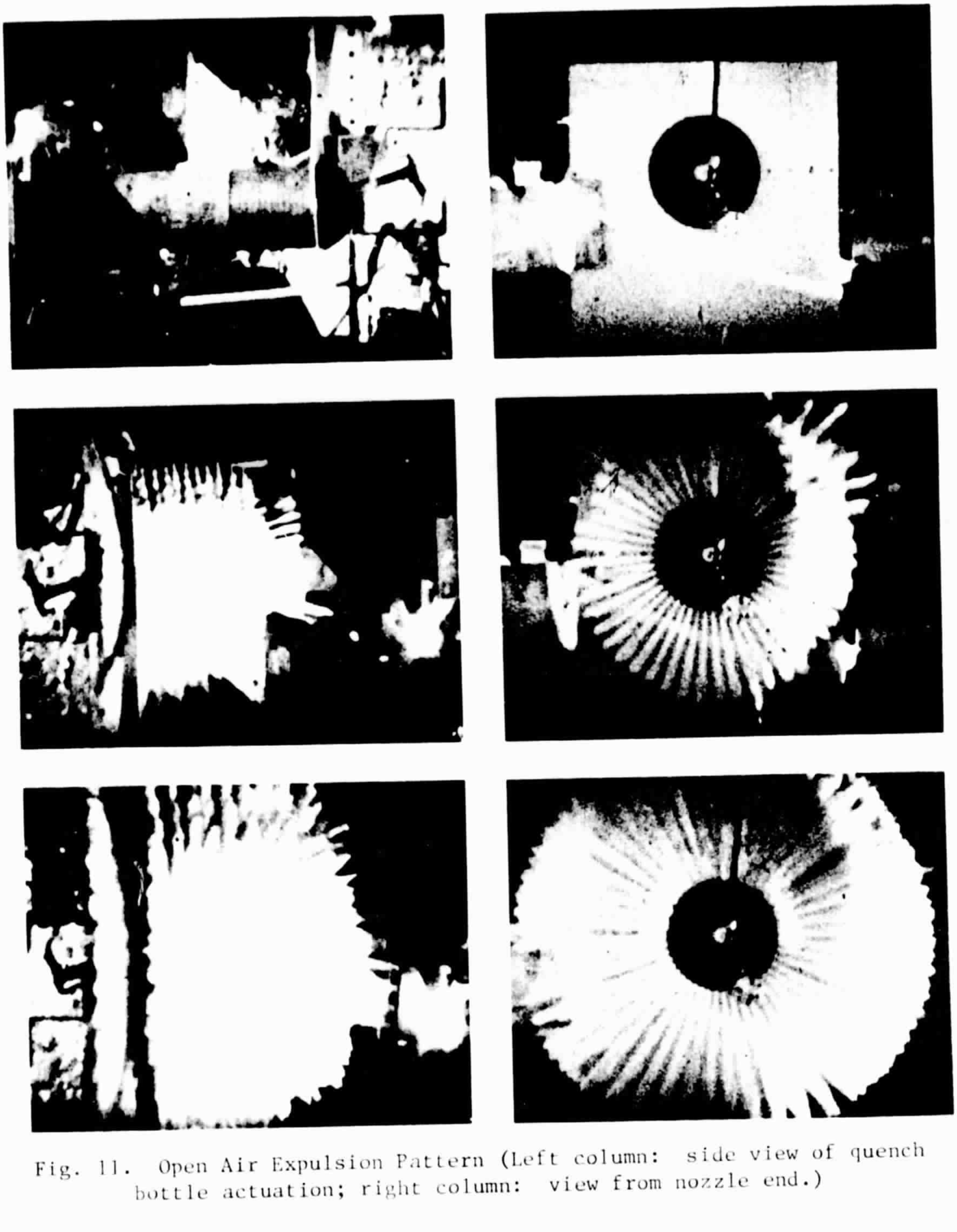


Fig. 11. Open Air Expulsion Pattern (Left column: side view of quench bottle actuation; right column: view from nozzle end.)

Table 2

QUENCH DEMONSTRATION TEST SUMMARY

OPERATING CONDITIONS	VID MOTORS SMALL SCALE	VID MOTORS FULL SCALE	VLN MOTORS SMALL SCALE	VLN MOTORS FULL SCALE
Chamber pressure (P_c), N/cm^2	351/176	263	407/193	411/91
L*				
First quench, cm	4267	4452	4928	5611
Second quench, cm	7112		8306	6566
Liquid used (Wt_{liq}/ burn surface), g/cm^2	0.3233	0.3374	0.3233	0.3726
Mass rate ($\dot{m}_{liq}/\dot{m}_{gas}$)	16 → 23	22	7 → 18	22 → 45
Burn duration, s	2.7 → 4.3	34 (first pulse)	1.3 → 1.6	39 (first pulse) 18 (second pulse)
Back pressure (P_a), N/cm^2	0.62	10.14	0.69	9.10
TEST RESULTS				
Successful quench[a]	3	1	2	2
Spontaneous reignition	No (3)	Yes (first pulse)[b]	No (2)	No (first pulse)[c] Yes (second pulse)

[a] No failures occurred.

[b] Post-quench termination ground cooling (fog) system failed to function as planned, spontaneous reignition occurred, and a successful second burn-to-depletion pulse was achieved.

[c] Active reignition was accomplished 8 s after first quench termination and prior to predicted time of auto-ignition.

Table 3

HEUS-RS FLIGHT-WEIGHT MOTOR PERFORMANCE ESTIMATES*

MOTOR DESIGN/ MATERIALS	CONFIGURATION	PMF†
Current	Two-burn	0.89
	Single-burn	0.91
Advanced	Two-burn	0.92
	Single-burn	0.94

* I_{sp} Vac (ϵ = 45) = 2970 m/s for all cases.

† Grain weight ≈ 1400 kg.

weight version of the present test weight design. This design is constrained by (1) use of very conventional (low-performance) materials, (2) continued use of test weight liquid quantities, (3) retention of the cold-gas pressurization system, and (4) retention of the present regressive-burn grain design. The regressiveness of the grain design results in limited acceleration loading of the payload but at the expense of a lower motor mass fraction. The single-burn version noted is merely the two-burn motor with the quench system and reignition igniters removed. The lower (advanced) design reflects improved propellant mass fraction performance by (1) utilization of advanced materials, (2) use of a solid warm-gas pressurization source, (3) reduced weights of quench fluid, and (4) providing a constant pressure type of grain design. The use of lighter nozzle expansion cone materials would probably result in selection of a higher expansion ratio with improved delivered specific impulse; however, this potential has been ignored in the impulse predictions noted. The incentives for upgrading the motor performance in the future would be predicated on mission payoffs.

CONCLUSION AND FUTURE WORK

Based on projected requirements for a solid rocket motor for auxiliary or kick stage augmentation of the Shuttle Space Tug and IUS, it would

appear that the motor and component technologies being developed for the APM and HEUS motors will generally satisfy those requirements, except in details to be determined as the studies of the Space Tug and IUS are completed and the implementation modes selected for the latter two concepts.

Future work on the APM motor will involve the test firing of several heavy-walled, test-weight but full-scale motors, followed by a test firing of the first flight-weight motor at ground, open-air conditions at the JPL Edwards Test Station. This will be accomplished in the first quarter of CY 1976, and the present technology phase of the work will be concluded by the firing of the second flight-weight motor at a suitable altitude-simulation facility during the first half of CY 1977.

A final full-scale, test-weight HEUS-RS motor will be test fired to verify the full two-burn, two-termination capability of the motor design. This final test will also be conducted under simulated altitude conditions, probably at the Arnold Engineering Development Center during the first half of CY 1976, to demonstrate that thrust termination can be sustained.

REFERENCES

1. M.J. Cork, J.M. Driver, and J. Wright, "Shuttle/IUS Performance for Planetary Missions," Paper No. AAS 75-273, presented at the Twenty First Annual Meeting of AAS, August 26, 27, 28, 1975, Denver, Colorado.

2. R.L. Stoller, Jet Propulsion Laboratory, Pasadena, California, private communication.

3. E. Kunz, Thiokol Chemical Corp., Elkton, Md., private communication.

4. W.A. Stephen, T.C. Warren, and J.M. Humphrey, Ballistic Missile Propellant Evaluation Test Motor System - Super BATES, Report No. AFRPL-TR-74-70, United Technology Center, Sunnyvale, Calif., November 25, 1974.

5. R.L. Bailey and W.M. Jensen, "Advanced Composite Solid Rocket Motor Chamber," JANNAF OSWG and SMBWG 1974 Meeting, CPIA Publication 253, July 1974, pp. 237-256.

6. J.I. Shafer, Solid-Propellant Motors for High-Incremental-Velocity Low-Acceleration Maneuvers in Space, Technical Memorandum 33-528, Jet Propulsion Laboratory, Pasadena, California, March 1, 1972.

7. R.L. Bailey and D.E. Carmichael, "Pyrolized Materials Nozzle," 1971 JANNAF Combined Propulsion Meeting, CPIA Publication 219, Vol. I, October 1971, pp. 153-167.

8. An Evaluation of Advanced Propulsion Systems for Upper-Stage Applications, Report No. DAC-58213, McDonnell Douglas Astronautics Company, Huntington Beach, California, July 1969.

9. HEUS-RS Applications Study, Report D2-116262-1, Vol. I, II, The Boeing Co., Seattle, Washington, June 1972.

10. D.J. Rohrbaugh, "Restartable Solid Motor Stage for Shuttle Applications," AIAA Paper No. 73-1240, AIAA/SAE 9th Propulsion Conference, Las Vegas, Nevada, November 5-7, 1973.

11. R.M. Evans, "Development of a Two-Burner, Dual Hydroquench Solid Propellant Motor (HEUS/RS)," JANNAF Solid Propulsion Conference, Anaheim, California, September 30 to October 2, 1975.

12. Phase I Final Report - Evaluation of a High-Energy Upper Stage Rocket Motor (U), Report No. 3363-03, Aerojet General Corporation, Sacramento, California, June 3, 1968. CONFIDENTIAL.

13. Final Report - High Energy Upper Stage Rocket Motor (U), Report No. H250-12-5-2, Volumes I, II, III, Hercules, Inc., Magna, Utah, August 1, 1968. CONFIDENTIAL.

14. Phase I Study Report for the High Energy Upper Stage Program (U), Report No. E119-68, Volumes I, II, Thiokol Chemical Corp., Elkton, Maryland, July 29, 1968. CONFIDENTIAL.

15. R.B. Wetherell, W.M. Bogart, and J.W. Behm, "Development of a High Energy Upper Stage Solid Propellant Rocket Motor with Extinguishment Capability (U)," 26th JANNAF Solid Propulsion Meeting, July 14-16, 1970, Washington, D.C., CPIA Publication 196, Vol. I, May 1970, pp. 405-424. CONFIDENTIAL.

16. Final Report Phase II, High Energy Upper Stage Motor Program (U), Report No. H250-12-6-7, Volume V Classified Data, Hercules Inc., Magna, Utah, July 1970. CONFIDENTIAL.

NASA—JPL—Coml., L.A., Calif.